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FLIGHT TESTING A DIGITAL FLIGHT CONTROL SYSTEM: ISSUES AND RESULTS

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ABSTRACT

The AFTI/F-16 Advanced Development Program, a joint USAF/USN/US Army/NASA effort with General Dynamics (GD), has modified an F-16A to be a testbed for evaluating new flight control related technologies. The program has ~~presently~~ completed its first phase of flight testing ~~at~~ NASA Dryden Flight Research Facility, Edwards AFB, CA. Some of the unique technologies being developed on this program are: a triplex digital fly-by-wire flight control system which operates asynchronously, an analog independent backup unit (IBU), eight separate digital task-tailored control laws, and six decoupled (six degrees-of-freedom) controller options. Included among these task-tailored modes are normal operation modes, air-to-air combat modes, and air-to-surface combat modes. One unique aspect of this program was the heavy involvement of the AFTI/F-16 Joint Test Force throughout the entire system development (pre-flight test) phase of this program. This forced early design consideration to be given to pilot-vehicle interface issues. Through the use of the GD Simulator, the test pilot became an integral part of the flight control law design. The AFTI/F-16 can be landed in possibly nine different sets of control laws including its normal digital mode, seven different sensor reconfiguration digital modes, and the analog IBU. Much concern surfaced prior to first flight as to how landable these different modes were; this resulted in all the landing modes being extensively tested on the GD Simulator, the Flight Dynamics Lab LAMARS, and the NT-33 Inflight Simulator. To date two of these modes, the normal mode and the IBU, have been flight tested on the AFTI/F-16 itself and the flight test results were different from any of the simulators' predicted results. This has raised several issues on the use of simulators to accurately represent today's highly augmented fighter aircraft. This paper will discuss several flight test issues, how they were resolved, and their effect especially on the aircraft handling qualities. Specific topics which will be discussed are: the IBU, the effect of the asynchronous computer operation and system redundancy management has on the flight control laws and flight testing, and some handling qualities problems with combination coupled/decoupled control laws.

INTRODUCTION

The AFTI/F-16 Advanced Development Program is primarily oriented to the development, integration, and evaluation of new flight control technologies. The testbed used in this program is an FSD F-16A (Figure 1). In this aircraft the quad redundant analog flight control computer system was replaced with a triply redundant digital flight control computer system using three BDX-930 digital processors and a triply redundant analog independent backup unit. New control surfaces were attached to the aircraft and usage of existing control surfaces was changed to provide more capability and flexibility in the flight control law design and to allow limited six degree-of-freedom decoupled motion capability. The surfaces added to the aircraft were two vertical chin canards, which were attached below the engine inlet, to provide enhanced directional force and moment control and to provide drag modulation capability. The surface whose usage was changed was trailing edge flaps. In the F-16 these two surfaces are only used for roll control and as normal flaps in landing. In the AFTI/F-16, they are used additionally in maneuvering flight to enhance the onset and control of normal acceleration and to provide longitudinal decoupled flight control. This paper directs its attention to three major design issues which had an impact on the first phase of the program. These issues which will be discussed in the next three sections of this paper are a direct result of these system modifications. This paper addresses each issue, the tradeoffs, the results, and its impact in flight testing.

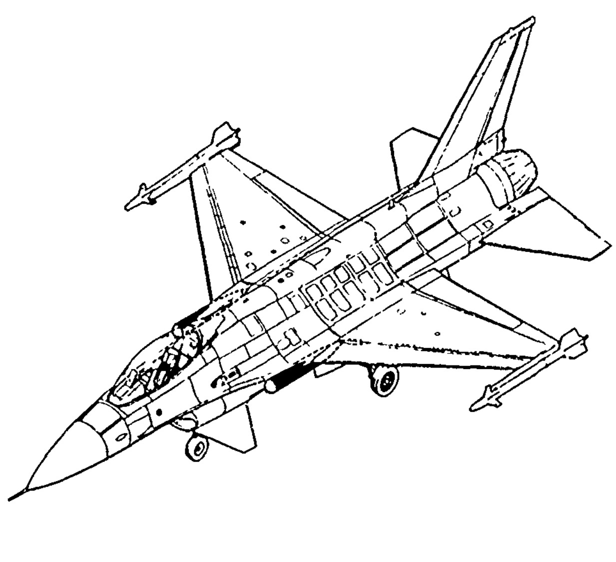

Figure 1 The AFTI/F-16

ISSUE: THE INDEPENDENT BACKUP UNIT (IBU)

The AFTI/F-16 digital flight control system was designed to meet several basic requirements:

1. Architecture: full authority, triplex digital, fly-by-wire system.

2. Reliability: DFCS failure rate resulting in loss of control of the aircraft will be less than 1 failure in every 10^7 flight hours.

3. Fail-Operation: the DFCS shall be fully operational (Operational State I as defined in MIL-F-9490D) after any first failure. After any second like failure, the control system will provide at least safe flight (Operational State III, MIL-F-9490D) with a probability of 0.95 being fully operational.

These requirements were to be met without reliance on an IBU. The primary reason for an IBU was to have a backup system which was independent of all the digital flight control software. This IBU was to be designed to allow for its removal after sufficient confidence was developed in the primary digital system. Also required was that the primary system and the IBU be designed so that the performance of the primary digital flight control system would not be affected by the removal of the IBU. The IBU alone would provide at least Level 3 flying qualities, as defined in MIL-F-8785B, throughout the flight envelope and at least Level 2 flying qualities in the landing phase.

Several pro/cons exist for having an analog independent backup for a digital flight control system. The basic reason for an IBU is to protect against unknown - unknowns in the digital software, especially generic software design errors. Due to an IBU being dissimilar redundancy, it also provides some protection against flight control EMI upsets. The presence of an IBU gives a definite increase in pilot and user - command confidence in flight testing a new digital flight control system. An IBU also increases the system loss-of-control reliability because it can survive some hardware failures.

Although the presence of an IBU has a lot of advantages, several disadvantages also exist. With an IBU, there will be increased system complexity which will result in increased cost. It will also be a source of additional flight control system failure points. The existance of an IBU can become a design "crutch" and be overrelied upon. An IBU can pose problems when there are no problems in the digital system: a nuisance automatic engagement and inadvertent or deliberate pilot engagements. The IBU may also require additional flight testing to clear its flight envelope. It might also mean additional pilot training to be proficient in the mode (assuming it is manually switchable from the cockpit).

To meet design requirements, the AFTI/F-16 program utilized a triplex analog IBU design. Since the digital flight control system is triplex, an analog card containing an IBU was located in each digital box. The IBU can be engaged through two methods. First a switch on the side-stick allowing the pilot to manually either engage or disengage the IBU. This gives the pilot the final authority to judge the health of the digital system. The second method of IBU engagement is by the digital system itself when it can no longer identify the last remaining good digital processor. In this case the IBU will automatically be engaged.

Early in the overall system design it was seen as beneficial to use the IBU when the redundancy management is unable to a high probability identify the last good remaining processor. The following scenario illustrates how the IBU can be used to improve the overall system safety. When no failure exists, the digital system uses the output of processor B to control the aircraft. (Processors A, B, and C are all compared to ensure B is good). If one processor has failed, the remaining two good processors will identify the sick processor and vote it off. The system will then use only the two good remaining processors. If a miscompare then occurs between the two remaining processors, they both go into self test with the anticipation that one will self test "GOOD" and the other "BAD". If this happens the aircraft flys home on the last good remaining processor (Figure 2). If both should test "GOOD", either processor could be chosen and safe operation should be assured. If both test "BAD" several options exist: one computer could be arbitrarily chosen (coin flip) to fly on, or the computer with the smallest output could be chosen (a small output is better than a hard over output). In any case, under these circumstances there is no guarantee of correctly choosing the last good remaining processor. On the AFTI/F-16 instead of arbitrarily choosing the last processor in this scenari , the system will automatically revert to IBU (Figure 3). Without an IBU, the redundancy management system would be forced to choose one of the last two processors to fly on. Some small risk exists that the incorrect processor would be chosen. The level of this risk is unknown since there are no known failures which would cause both processors to self test "BAD". This is clearly a case where the IBU is protecting against unknown - unknowns. Because this was a new flight control concept being developed and flight tested (digital, triplex, asynchronous design), it was felt to be more prudent to utilize the IBU. It should be noted that the reliability rate for loss-of-control (less than once every 10^7 flight hours) is still met even if the IBU is not in use in the above scenario.

From a control law standpoint, it was decided to make the IBU as simple as possible to keep the analog real estate small and keep the IBU as independent from hardware failures as possible. Under this criteria, the IBU design resulted in a system with three input paths (pitch, roll, yaw) and a single feedback path: pitch rate feedback which was the minimum necessary to maintain aircraft stability. In the roll and yaw axis no feedback was used even though the dutch roll characteristics were rather poor. The design resulted in a single gain system which had to have sufficient stability margins to be stable and still flyable from Mach 1.6 through touchdown speed. The critical region in determining the value of this gain was the low altitude/supersonic region. This resulted in forcing the gain to be low which caused the pitch damping at power approach airspeed to be low.

Prior to first flight of the AFTI/F-16 the IBU (named Original IBU) was evaluated in a power approach (PA) configuration on the NT-33A Inflight Simulator. While performing a PA task this IBU was given Level 3 flying qualities ratings with Cooper-Harper ratings of 8 to 9.5 in pitch and 4 to 5 in roll. The level 3 ratings were due to heavy, sluggish pitch response, a PIO tendency in high gain tasks, and heavy and extreme gust sensitivity in roll. At this point is was obvious a single gain IBU was not going to be sufficient to safely fly the aircraft throughout the entire flight envelope and still safely land it.

As an interim fix, the single gain system was adjusted to give much improved low speed dynamics but had to be placarded against supersonic flight because of insufficient stability margins (named First Flight IBU). (Figure 4). This IBU doubled the forward path proportional gain. It was to be used on the aircraft until the final IBU (named Block III IBU) could be built and incorporated into each digital processor box. The Block III IBU was a dual gain IBU having a low gain to accommodate the supersonic stability margains and upon lowering the gear handle, a high gain for landing. Both these designs were then tested on a second series of NT-33A tests and both found to have good level 2 handling qualities.

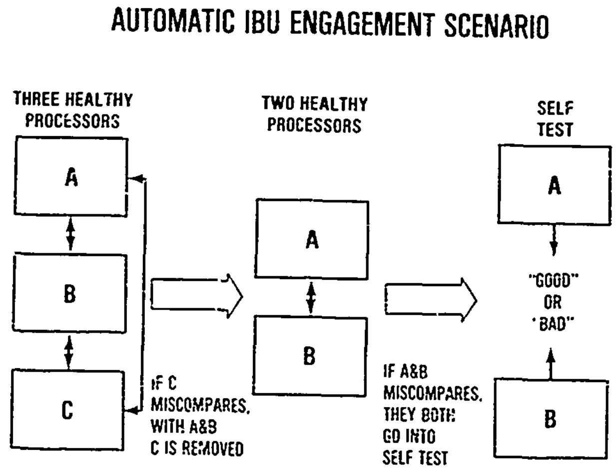

Figure 2 Failure Scenario

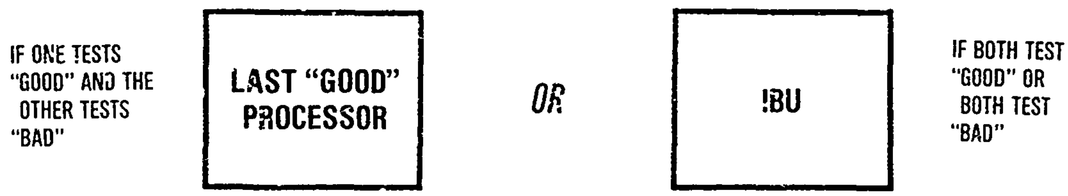

Figure 3 Result of Self Test

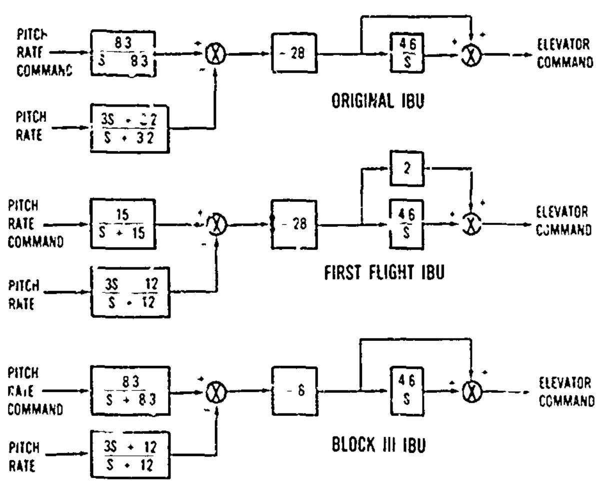

Figure 4 IBU Development History

As first flight date of the AFTI/F-16 approached, a major issue began to surface with respect to the intentional inflight usage of the IBU. Should the IBU be intentionally engaged inflight, how often, and in what flight conditions? The flight test community was definitely in favor of testing the IBU inflight and desired it very early in the program (second or third flight). The advantages of doing this would be to gain confidence in the IBU under controlled rather than last chance emergency conditions. Under this scenario, the IBU would be tested under benign flight conditions and there would be a healthy digital flight control system to immediately revert back to if the IBU flying qualities showed to be unsatisfactory. Since the NT-33A evaluated the flying qualities to be very close to Level 1, this did not seem to be too likely. From a redundancy/reliability standpoint the IBU is monitored for failure at all times, even while flying in a digital flight control mode. The IBU on the AFTI/F-16, being a triplex system, is fail-operative after first failure but has no protection against a second failure. The IBU hardware has a predicted mission reliability of less than 7X 10^{-6}failure per flight hour.

Several reasons also exist for not intentionally turning on the IBU. The IBU is an emergency mode and was deliberately designed to be "bare bones - get home and land" mode. It has no lateral - directional feedbacks nor aileron - rudder interconnect nor yaw damper; thus it was expected to have poor lateral - directional flying qualities. In any case, the flying qualities are worse than the normal digital modes. Also, if for some reason the system would not reset back to its digital normal mode, a real emergency would exist.

Historical data has not shown a very good track record concerning inflight engagements of flight control system backups. Aircraft such as the B-47, B-66, Tornado, and Concorde have mechanical backup systems for their primary flight control systems. In the B-47 and B-66 these mechanical systems were maintenance nightmares and were many times not well maintained. In the B-66, twice pilots switched to the mechanical system and found it inoperative. Neither pilot was able to switch back to the hydraulic (primary) system which resulted in loss of both aircraft.

Those in favor of engagement of the IBU inflight won and the IBU was first engaged during the third flight of the aircraft. All the pilots commented that the IBU had degraded flying qualities as compared to the Standard Normal Mode but that the flying qualities were sufficient as a backup mode. Pitch axis was very stable but possessed moderate to heavy stick forces. It had a lightly damped dutch roll which was excited by roll or yaw inputs. The IBU to date has been flown out to Mach 1.2. At that speed the pilot got into a lateral PIO once when he excited the dutch roll with a maximum rudder input.

The IBU is now routinely engaged during the course of flight testing. As the flight envelope is expanded, the IBU is now one of the first modes to be evaluated at each new flight condition. The philosophy is that the IBU will be tested for safe operations to give confidence in case of non-resettable automatic IBU engagement.

ISSUE: CONTROL LAW AND REDUNDANCY MANAGEMENT CO-EXISTING IN A DIGITAL ENVIRONMENT

A major objective of the AFTI/F-16 program was to develop and evaluate new concepts in flight control design. One of the major concepts being developed was that of multimode design containing task-tailored control laws. This implies a separate flight control mode be designed for each type of combat task to be flown (bombing, air-to-air gunnery, etc.). Also required with this control law structure was limited authority six degree-of-freedom decoupled set of flight control laws. Additionally required was separate reconfiguration modes which allow for continued flight after the loss of flight control sensors (pitch rate, roll rate, tec.). In other words, a very complex set of control laws was required to be developed. A second major concept being developed was a triplex, digital flight control computer system. This system was required to utilize software to the greatest extent possible to perform all control law and all redundancy management functions. Early in this program, some important decisions were made concerning the approach used to implement these concepts. At the time these decisions were made, it was felt their effect would not impact each other. As the design progressed from development to mechanization to flight testing, it became obvious these decisions were greatly intertwined and system adjustments were necessary to allow for harmonious system operation. One of these decisions was to operate the digital computers asynchronously with respect to time. The other decision was to design the control laws utilizing Linear Quadratic Synthesis (LQS).

Asynchronous digital computer operation implies that the individual clocks in each processor will operate independent of each other thus implying the time skew between each processor will not be controlled (Figure 5). Therefore each processor will receive its input data at different times and will complete the output surface computations at different times. Prior to making the decision to go asynchronous, a trade study was performed. Some of the main conclusions of this study were:

Synchronous Operation

Advantages:

1. Simpler Operational Flight Program (OFP) verification and testing.

2. Cross-channel monitor trip level at output selector/monitor plane can be set to a near zero value (a cross-channel difference can be used as a failure indication).

Disadvantages:

1. Sync function must be carefully designed so as not to introduce a single point failure possibility into the system.

2. Design is not inherently fault-tolerant. Unless special care is taken, a transient condition in one branch will in general result in a branch being temporarily disconnected.

Asynchronous Operation

Advantages:

1. More fault-tolerant since branches are not expected to be in exact agreement.

2. Insensitivity to short term electromagnetic interference effects is enhanced since data which is modified or in error is likely to be sensed in diverse portions of the redundancy management function.

Disadvantages:

1. Increased data acquisition speed requirement to prevent dynamic responses from being identified as faults at the output selector/monitor plane.

2. Exact skew conditions are difficult to repeat to obtain identical results during testing.

To operate the computers asynchronously was itself a new concept never before tried in an aircraft flight control system. As a result of the trade study, it was decided to proceed with asynchronous computer operation even though this was obviously riskier than taking the more conventional approach of synchronous operation.

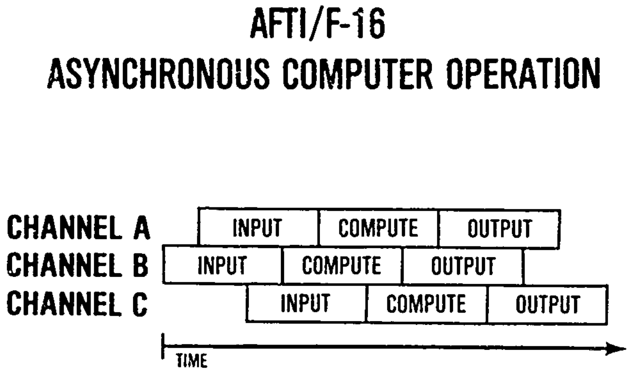

Figure 5 Time Skew

An equally risky approach was taken in using the LQS approach to the control law design. The control laws were developed for many point conditions in a batch computer mode utilizing a digital computer program called DIGICON. The goal in the control law design was to have a very quick responding system, both in pitch rate and onset of normal acceleration, and this system would provide gust alleviation and be relatively invariant to uncertainties in the airframe aerodynamic derivatives. To meet these design goals, a multi-state feedback system was designed which had very high gains on the forward path signals, especially the error signals between commanded response and actual aircraft response. The result of these high gains was a system that has very large amplitude and high frequency content in the output signal to the control surface actuators.

Within the digital flight control system, two major tasks are being performed during each frame: the flight control law computations and the redundancy management computations. Initial development of these two systems (control law versus redundancy management) was done separately; it was not until later in the preflight development phase of the program that the two systems were integrated together.

The redundancy management system is an integral part of any multiple computer system. The purpose of the redundancy management system is to ensure that the aircraft is always being flown by a healthy processor (s). One component of this system which highly interacts with the control laws is the output selector/monitor (S/M). The function of the output S/M is to compare the surface commands of all three processors to detect computational failures. Each processor has its own output S/M which compares all eight of its surface command outputs with those of the other two processors. If any processor's output (including itself) differs by more than a given percentage from the other processors, that computer output from that processor is identified as having possibly failed. If this condition persists for seven computational frames, that output from that processor is voted off-line. This given percentage is called the trip level and its purpose is to prevent a sick output of any processor from commanding an aircraft surface. In order to have early warning of a possible failure and prevent any aircraft failure transients, ideally this trip level would be set near zero percent difference. But this is impossible because of asynchronous nature of the system. This asynchronism will allow each processor to be time skewed from each other processor - each processor's output being different even for a perfectly healthy system (Figure 6).

The difficulty arises in determining an acceptable trip level which allows normal operation to continue in a time skewed environment and still identify output failures at a safe level. For a healthy system, three factors will affect the inter-channel difference between the outputs from the three processors: the time skew between the three processors, the change of the input signals (both commands and sensor inputs) in that time, and the control law gains and structure which amplify these differences. As stated above the desired size of this trip level is controlled by two opposing factors. The trip level must be large enough to allow for a normal interchannel difference due to the three factors listed above so it will not erroneously declare an output failure. Opposing this, the trip level must be small enough such that a real failure can be identified before it can produce a large (unsafe) aircraft transient when switching from a sick output to a healthy output. Although this non-zero trip level allows the system to be fault tolerant (it could possibly allow a small short term transient to pass through system without declaring a failure), a deficiency of this asynchronous system is its inability to always distinguish the difference between a time skewed output mis-compare and an actual failed output at a low threshold level.

When the total system was integrated initially the inability of the redundancy management system and the control laws to work in perfect harmony became obvious. Originally the output trip level was set at a constant value of 15% of full scale deflection of each given surface. When the total system was first tested on the simulator, it was found that large inputs, especially at moderate to high frequency, were exceeding this 15% trip level in the output S/M. At this point comparisons were made between the AFTI/F-16 control laws and those in the F-16. It was found in some cases that the gains in the AFTI/F-16 control laws were sometimes many times larger than equivalent gains in the F-16. Some suggested the AFTI/F-16 gains were unrealistically high. In any case these high gains generated by the LQS approach proved to be a real deficiency of that type of control law design.

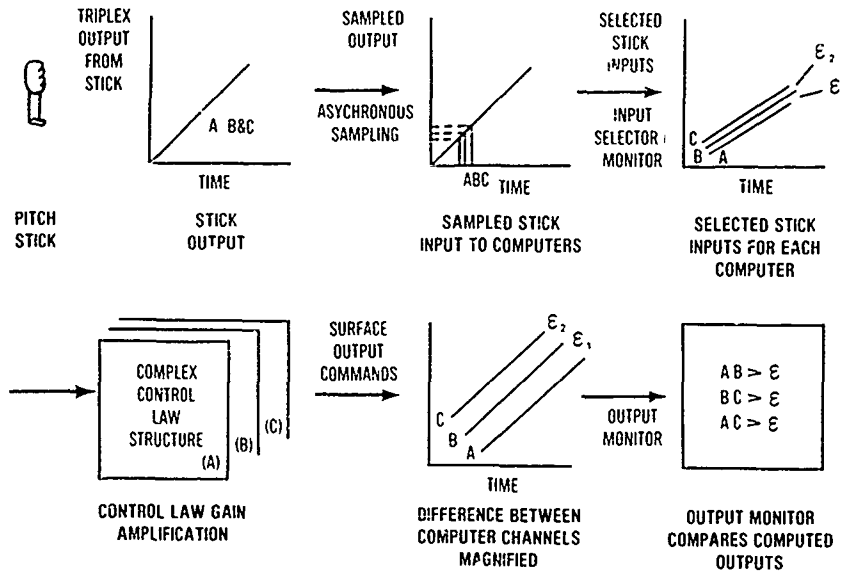

Figure 6 System Interaction

The LQS design methodology used on the AFTI/F-16 varied the gains in the forward path and feedback paths to satisfy a cost function and performance criteria. Unfortunately the method did not give the designer a sensitivity analysis of the effect of the gain on system performance. The end effect was the final design resulted in some very large gains which could have been reduced at a very small loss of performance. These large gains caused a lot of problems when integrated with entire system. They would produce a large amount of control surface activity which meant hinge moment limits were a bigger problem, more surface rate limiting, and large demands on the hydraulic system. The end result with respect to the redundancy management was that for maximum processor skews, a very small difference in input signal magnitude could result in very large differences at the output plane for non-failed conditions. Ultimately the gains had to be reduced to be more realistic and to coexist with the redundancy management. The lowering of the gains was not done without some penality; in some cases it resulted in reduced robustness of the aircraft response and possible degradation of aircraft handling qualities. The primary area this gain reduction occurred was the forward path to the elevator in all the flight control modes and the forward path to the trailing edge flaps in the decoupled control modes. The region of the flight envelope most affected by the gain reduction was the high altitude, low dynamic pressure areas. In this region, the sluggish performance of the basic airframe is masked by the very high gain control laws to achieve good tracking performance and improved handling qualities; therefore, the gain reduction had its greatest effect there. In general though, gains were reduced throughout the flight envelope. The effect of reducing the gains in the forward path to the elevator was a reduction of system bandwidth in the pitch response. The effect of reducing the gains in the forward path to the elevator was a reduction of system bandwidth in the pitch response. The effect of reducing the gains in the forward path to the trailing edge flaps was an increase in the impurity present in the decoupled control options. The end result was the control laws had to be changed from their optimal design to live in harmony with the redundancy management system.

The redundancy management system was also changed to be more compatible with the control laws. The constant 15% trip level was changed to be a variable trip level based on the rate of change of a specific output in its own processor. In most cases the trip level is 15% but it can reach as high as 30%. This change still provides the same level of protection against a real failure. If one processor has a failure forcing one of its outputs to increase at a high rate (thus increasing its trip level to 30%) the other two processors may still be using 15% for that output, thus voting it as failed as early as if all three processors were using 15%. Other changes also made to the redundancy management system because of the high gain control laws were to increase the rate at which some inputs are sampled and to increase the update rate of gain tables. These changes were necessary to prevent large, rapid spikes to propagate to the output S/M plane. Therefore, the end result was the redundancy management system also had to be modified to live in harmony with the control laws. This system adjustment, to make the control laws and redundancy management system work in harmony, was primarily performed on the simulator. During the flight testing phase (a total of 118 flights, 177 flight hours) only one inflight flight control system fault indication occurr d which resulted in a control law gain adjustment.

As a result of the flight test program, several conclusions can be made concerning the operation of high gain control laws in an asynchronous computer environment. First (and most important), these two systems can be made to work together successfully. During the last two weeks of flight testing, the pilot was permitted to aggressively fly air-to-air and air-to-ground combat scenarios and the system performed flawlessly with zero flight control problems. The second conclusion is that there is an interdependency between the control laws and the redundancy management in an asynchronous computer system, and the two parts cannot be developed independent of one another. The smaller the control laws gains are or the faster the computer frame rate is, the less this will be a problem. On the AFTI/F-16, the computer processors have a frame length of approximately 16 milliseconds which allows the worst case skewing

between any two of the three processors to be about 8 milliseconds. For this system, there obviously was a limit to how large the control law gains could be and still coexist with the redundancy management system. With these gain restrictions, the control laws on the AFTI/F-16 were still sufficiently robust that they received Level 1 Cooper-Harper Handling Qualities Ratings in all the combat modes. These control laws were also determined to have improved handling qualities over the basic F-16 in all combat modes and in power approach and landing. Third, this asynchronous design proved to be very fault tolerant. During the flight testing phase, forty-one inflight flight control system fault indications occurred. The effect of these fault indications ranged from no loss of the system redundancy up to the loss of two channels of redundancy (i.e., flying on only one processor). In almost all these cases the system was resettable to a zero fault condition. In no case was there a degradation or change in the aircraft handling qualities due to a fault indication.

ISSUE: DECOUPLED CONTROL LAWS

The AFTI/F-16 digital flight control system contains eight primary task-tailored control law modes (Figure 7). These modes are full authority and are optimized for specific tasks such as air-to-air gunnery and air-to-surface bombing. For verrier, fine tracking adjustments, six decoupled control options were developed. These options include: in the longitudinal axis - pitch pointing, direct lift, and vertical translation (Figure 8); in the lateral - directional axis - yaw pointing, direct side force (also named wings level turn or flat turn), and lateral translation (Figure 9). (See Reference 1 for a description of these modes). These decoupled control options are superimposed on the primary control laws as a secondary means of precisely tracking a target in the final stages of a tracking solution.

AFTI/F-16 Multimode Flight Controller Commands

CONTROLLER	MULITMODE			
	STD NORMAL	*STD BOMBING*	*STD ASG*	*STD AAG*
SIDE STICK (Pitch)	A_N COMMAND	A_N COMMAND	Q COMMAND	Q COMMAND
SIDE STICK (Roll)	ROLL RATE COM	ROLL RATE COM	ROLL RATE COM	ROLL RATE COM
RUDDER PEDAL	RUDDER DEFLECTION	FLAT TURN	FLAT TURN	FLAT TURN
THROTTLE (Twist)	NONE	NONE	NONE	NONE
	DECOUPLED	*DECOUPLED*	*DECOUPLED*	*DECOUPLED*
SIDE STICK (Pitch)	FPME*	FPME	PRME*	PRME
SIDE STICK (Roll)	ROLL RATE COM	ROLL RATE COM	ROLL RATE COM	ROLL RATE COM
RUDDER PEDAL	TRANSLATION	FLAT TURN	POINTING	POINTING
THROTTLE (Twist)	TRANSLATION	DIRECT LIFT	POINTING	POINTING

**FPME – FLIGHT PATH MANEUVER ENHANCEMENT*
PRME – PITCH RATE MANEUVER ENHANCEMENT

Figure 7 AFTI/F-16 Flight Control Modes

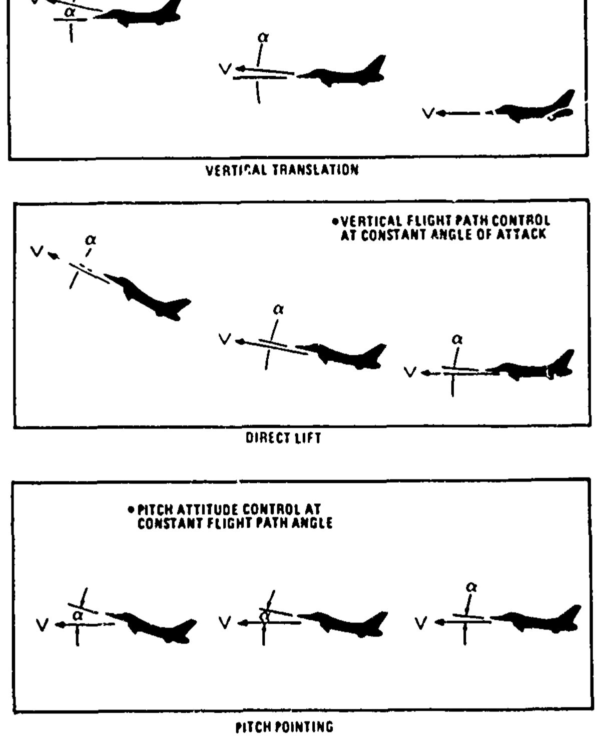

Figure 8 Longitudinal Decoupled Options

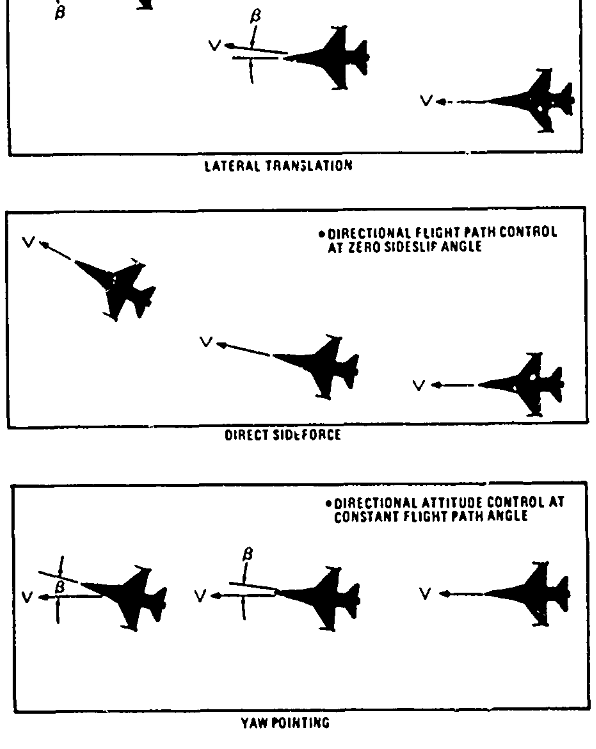

Figure 9 Lat-Dir Decoupled Options

During the Fighter CCV program (utilizing a YF-16 as a testbed) all six of the decoupled modes were evaluated inflight. On that aircraft, the decoupled control responses were generated in an auxiliary flight control computer which was not an integral part of the primary analog quad redundant flight control system. This was to allow the aircraft to always, instantly revert back to its primary coupled control laws if the aircraft got into a problem area during the decoupled operation. Since the baseline YF-16 feedback remained intact within the primary computer, specific feedbacks not desired for decoupled operation were cancelled by predicted open loop response signals from the auxiliary computer. Gain scheduling as a function of air data parameters provided operation of the decoupled options over a wide range of flight conditions.

In the AFTI/F-16 flight control system, the entire control law architecture was redesigned from the existingF-16 control laws. There was no requirement (nor need) to jury-rig the decoupled control system to an existing full authority control system as was done in the Fighter CCV program. Therefore two methods existed for integrating the limited authority decoupled control laws with the full authority system. The first method was the open loop approach similar to the Fighter CCV; the second method was the closed loop approach integrating the decoupled control laws with the entire control law structure and utilizing the multiple sensor feedbacks that were available. Being an advanced development program evaluating new aspects of integrated flight control technology, the latter approach was chosen.

In the AFTI/F-16 program, the first major decision to be made with respect to decoupled control was what type of controllers would be used to input decoupled commands. On the Fighter CCV, a miniature two-axis force controller was installed on top of the YF-16 side stick controller for commanding the decoupled modes. As a pilot option, the rudder pedals could be used to input lateral-directional modes. The pilots found that this two-axis controller produced a lot of crosstalk with the coupled controller anytime a decoupled input was made. In other words, it was difficult to make a decoupled input without unintentionally deflecting the coupled (primary) side stick controller. The AFTI/F-16 chose to use separate controllers (not co-located) to command decoupled inputs. This was to prevent cross-talk or interference between controllers. Only two controllers were necessary to make all decoupled inputs. The pilot controller chosen for lateral-directional inputs was the rudder pedals. Flight testing showed the rudder pedals to be very natural for this task. For the longitudinal inputs, the throttle grip was modified to have a dual function as a throttle and as a decoupled motion controller. As a controller the throttle is twisted aft to command up motion and twisted forward to command down motion (Figure 10). Flight testing showed the twist throttle to have several problems. The pilot tended to put inadvertant twist throttle inputs in during high gain tracking tasks and high G-loading. If the twist throttle was held slightly out of detent, the trailing edge flaps would integrate to their limits, hence greatly increasing drag. Sometimes the pilot's first indication of an inadvertant input was the slowing of the aircraft or the onset of wing buffet. The second problem is the twist grip's harmony with the side stick controller. The pilots found it difficult to use the twist throttle and the pitch stick simultaneously to control the pitch axis. To use the throttle, the pilot generally had to freeze the pitch stick which tended to increase pilots overall workload.

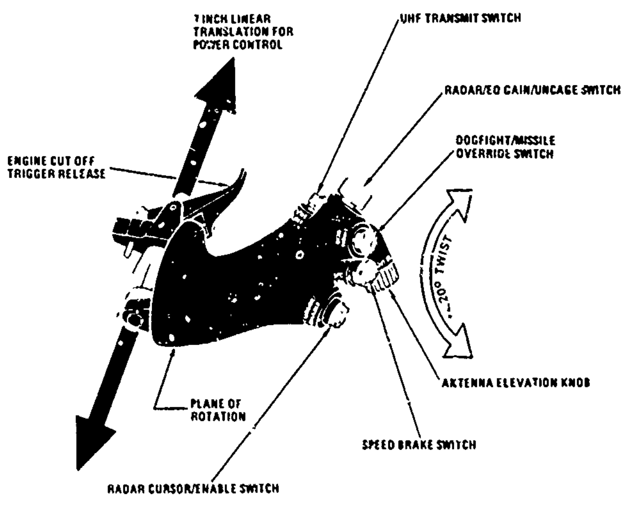

Figure 10 Throttle Twist Grip

The utility of the decoupled control options is very dependent on their response being linear, predictable, free of large impurities, relatively quick, and of sufficient magnitude to be useful. It is not necessary for these control laws to be able to perform gross acquisition tasks, but they must be very effective at vernier tracking tasks to be useful. Results from the Fighter CCV program showed the highest payoff for these modes to be: pitch and yaw pointing for gunnery tasks; direct side force for air-to-air tracking, strafing and bombing; direct lift for defensive maneuvering to confuse an attacker; and lateral translation for strafing or landing in a crosswind. Vertical translation showed very little utility for any task.

Below is a summary of results from flight testing the decoupled control options on the AFTI/F-16. This summary includes what each controller commanded, authority levels used by the pilots, and the option's primary utility.

- For flat turn, the rudder pedal commands an acceleration. The average maximum authority used by the pilots was 0.5 G's (Figure 11). Of the six modes evaluated, this mode was found to be the most useful for reducing the time to a firing solution on a combat target. Flat turn was found ideal in air-to-surface

bombing and strafing to null small tracking errors on the target. Approximately 3 degrees (50 milliradians) was the changeover point where conventional turning was quicker than flat turn. The pilots found flat turn was especially optimum in eliminating 10 milliradian errors; for removing large tracking errors the lateral accelerations produced by the flat turn (0.7 to 1.0 G's side force) was objectionable to the pilots. For these large errors, the pilots found conventional banking was best for the acquisition task and flat turn best for removing the final tracking error.

- For direct lift, the twist throttle commands an acceleration. The average maximum authority used by the pilots was 0.7 G's. Although the response to the twist throttle input was smooth, linear, and predictable, direct lift did not show great utility since the pitch stick also provided precise flight path control for air-to-ground tasks.

- For pitch and yaw pointing, the twist throttle and rudder pedal command an angular rate. The average maximum authority used for pitch pointing was 3.0 degrees; yaw pointing was 3.5 degrees. The pointing modes were initially programmed to be used in air-to-air combat and strafing. But after some initial flight testing, the pilots discovered they preferred controlling flight path (flat turn, direct lift) rather than weapon line pointing for the strafing task. Pitch pointing was found useful in tracking a cooperative air-to-air target aircraft, but for a jinking target, pitch pointing's utility greatly diminished because of its limited authority and speed. As with direct lift, the pitch stick could track as well as pitch pointing thereby further diminishing its need. Yaw pointing's utility was slightly better, but it was only good for small lateral corrections. When maximum pointing angles were commanded, roll coupling was sometimes a problem. All the pilots commented they would have preferred commanding pointing angle rather than pointing rate for both pointing modes.

- For vertical and lateral translation, the twist throttle and rudder pedal commands an acceleration. In close formation flying the pilots felt these modes actually increased their workload over conventional techniques. This was possibly due to these modes being acceleration command systems which forced the pilot to provide lead compensation to precisely position the aircraft. All the pilots stated the modes would have had more utility if they were velocity command systems.

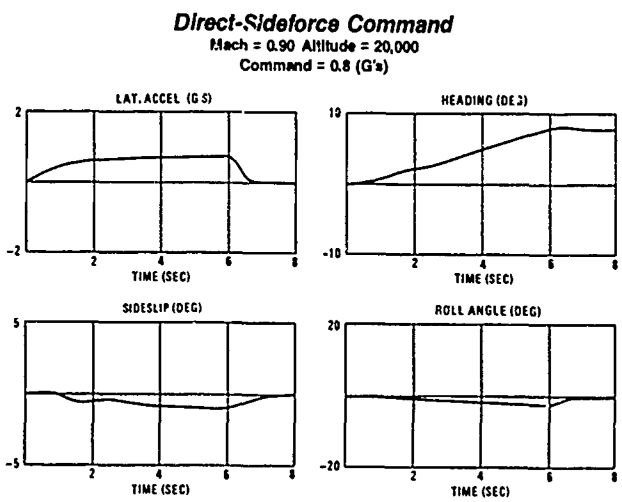

Figure 11 Flat Turn Time History

Purity of the decoupled responses became an issue as the mode designs began to finalize. With the open loop system used on the Fighter CCV, the purity level was a function of how accurately the designer could predict the aerodynamic forces produced by each control surface. In the closed loop system, the designer's ability to feedback the correct variables will provide the primary influence on mode purity. As an example, in the direct side force mode the rudder pedal input is a lateral acceleration command. The lateral acceleration error passes a proportional plus integral network then commands a canard surface deflection. To maintain zero sideslip angle, sideslip rate feedback is passed through a proportional plus integral network then commands a rudder surface deflection. The difficulty occurs in the ability to accurately measure (or calculate) sideslip rate. The ideal way to calculate sideslip rate would be to use sideslip angle. Unfortunately, no location on the aircraft could be found from which a sideslip angle probe worked accurately at all flight conditions. As a result, sideslip rate was calculated by using yaw rate, lateral acceleration, and a small roll rate component to compensate for any angle-of-attack. Without there being sideslip angle feedback, any steady state sideslip that exists at the beginning of a direct side force maneuver will never be washed out during the maneuver. Also for very slow command inputs, the yaw rate can be so low that the yaw rate sensor is ineffective in measuring it, thus, sideslip angular error will build up.

Flight testing, though, showed that the mode purity was not nearly as critical as originally thought. There appeared to be a purity threshold above which further improvement had no effect on pilot acceptability or task performance. For pitch pointing and yaw pointing modes, the impurities were generally proverse in the form of acceleration in the direction of the pointing angle. It seemed that the impurities could be fairly large, especially air-to-air, and have no effect on pilot performance. During strafing, though, the down acceleration impurity associated with down pitch pointing was rather objectionable. For vertical and lateral translation, the impurities were in the form of aircraft rotation and the amount of acceptable impurity was very task dependent. If the task was formation flying or aerial refueling, a little impurity can make the pilot very nervous. Any other tasks where collision avoidance is not an issue, the impurities were not important. Proverse rotational impurity was less disorienting than adverse. For direct lift,

purity did not appear to be very important. For flat turn, the impurity was in the form of sideslip angle. Proverse sideslip, because it is in the direction of turn, is less disconcerting to the pilot than adverse sideslip. From a structural standpoint, proverse sideslip during a flat turn can generate large loads on the vertical tail. Therefore, flat turn purity may be primarily dictated by structural strength rather than pilot performance.

These were just a few of the problems encountered while trying to integrate decoupled/coupled flight control laws into one system. This program has demonstrated that decoupled control laws, especially flat turn, can be used to improve weapon system effectiveness.

CONCLUSIONS

As a result of these design issues, several conclusions can be made:

1. An IBU is effective in improving user confidence while flight testing a new complex digital flight control system. An IBU also provides a safeguard against generic software failures.

2. An asynchronous computer system works; the redundancy management system for asynchronous operation can live in harmony with high gain control laws but their designs will be interdependent on one another. This asynchronous computer system (as tested on the AFTI/F-16) is a highly fault tolerant system.

3. Of the six decoupled control law options tested, flat turn is the most effective in reducing time-to-null weapon line tracking error relative to conventional tracking methods. The rudder pedals are ideal for controlling flat turn and 0.5 G's is its optimum authority limit.

EPILOGUE

The AFTI/F-16, after completion of all system modifications, began its flight test phase in July 1982. The flight testing was successfully completed in July 1983 at which time Phase I of the program (Digital Flight Control System Phase) was concluded and aircraft modifications for Phase II of the program (Automated Maneuvering Attack System Phase) began. In this second phase of the program, the aircraft will be tested in a much harsher environment of low altitude, automatic weapon delivery. In this environment good flight control system reliability and high pilot confidence in the system is essential to successfully achieve program goals. The ability o the aircraft to maneuver precisely and aggressively and to accurately deliver weapons in a high-G environment is also an important factor in the second phase of this program. To allow the AFTI/F-16 program to reach its goal these issues and many others had to be (and will have to be) successfully resolved.

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